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OCT 77 J WEISS, J SOCHACKI

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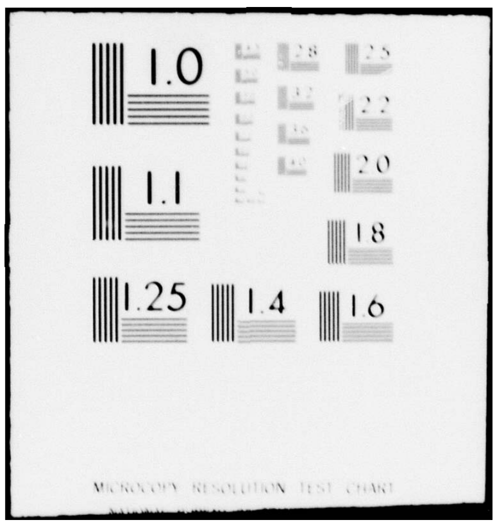
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METHOD FOR DETERMINATION OF THE COEFFICIENTS OF GASODYNAMIC AND HEAT LOSSES IN SOLID PROPELLANT ROCKET ENGINES

by

J. Weiss and J. Sochacki

FOREIGN TECHNOLOGY DIVISION

FTD -ID(RS)T-1746-77

EDITED TRANSLATION

FTD-ID(RS)T-1746-77 26 October 1977

MICROFICHE NR: FTD-77-C-001342

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By: J. Weiss and J. Sochacki

English pages: 11

Source: Archiwum Termodynamiki i Spalania, Warsaw, Vol. 6, Nr. 3, March 1975, pp 367-372

Country of origin: Poland
Translated by: LINGUISTIC SYSTEMS, INC.
F33657-76-D-0389
F. Saleski
Requester: AFRPL
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Method for Determination of the Coefficients of Gasodynamic and Heat Losses in Solid Propellant Rocket Engines

J. Weiss, J. Sochacki

A method is presented for the experimental determination of the coefficients of gasodynamic loss connected with gas flow through a nozzle and of heat losses occurring during combustion of the propellant charge.

The dependences describing the heat and gasodynamic processes flowing through solid propellant rocket engines are usually established for an ideal system. In practice when establishing rocket characteristics it is necessary to consider the following properties of a real working process.

The absence of parallel gas fluxes flowing through the nozzle. This factor is usually negligible when introducing a formula for propulsive force. In actuality in the divergent part of the nozzle the gas flux takes on a conic shape and then only the inner portion of the flux shifts in parallel to the axis of the nozzle, however, the remaining one is deflected at a certain angle with respect to the axis. This leads to a decrease in propulsive force.

The existence of friction between the gas and the walls of the combustion chamber and the nozzle as well as the heterogeneous distribution of gas parameters in a cross-section of the flux. These phenomena are conditioned by the qualities of actual gas and engine construction. They also cause a decrease in propulsive force.

Decrease in the passage section of the nozzle caused by the emergence of the flux boundary layer. A decrease in the jet's critical area causes a change in the second output of gas, however an decrease in the exhaust area influences a decrease of the degree of gas expansion and likewise lowers the value of rocket propulsion.

Heat losses occurring during fuel combustion. As a result of heat transmission through hot gases to the walls of the combustion chamber energetic losses occur. For that reason the actual running determined by the propellant force $f_0 = RT_0$ is less compared with that determined by way of theoretical calculations. Heat losses have an uneven pattern.. At the beginning of the engine run they have the greatest value but decrease within the limits of the combustion charge.

Pressure drop along the combustion chamber. The value of the pressure drop depends chiefly on the relation between the critical

jet area and the chamber passage area and several structural qualities of the engine. In the chamber the appearance of local obstacles, e.g., a grate, influences the pressure drop to a considerable degree. The pressure drop should also be considered when calculating the force of propulsion.

In the theory of solid propellant rocket engines the above factors are taken into consideration with the aid of correction factors introduced to corresponding conditions describing gas exhaust. By using coefficients of gasodynamic and heat losses conformity of the results of theoretical calculations with the experimental data is achieved.

Gasodynamic losses are determined with the help of the coefficients of losses of gas flux velocity φ_1, and the unevenness of real output φ_2.

According to the data in the literature, for typical engines, the values of these coefficients are formed within rather broad limits.

$$\varphi_1 = 0.9\text{-}0.97; \qquad \varphi_2 = 0.85\text{-}0.98$$

When considering heat losses in practical caluclations the most often used is the empirical dependence

$$\chi = 1 - \frac{a}{1 + b\psi}$$

where a and b are the constants depending on the type of charge

ψ is the relative part of the burned propellant

For example for engines without thermally insulated walls, with a tubular charge having flank combustion and relatively short running periods on the average $\chi = 0.9$ is accepted.

Noting the rather wide scope of changes in these quantities, the elaboration of a general method of experimental evaluation of correction factors for a specific type of engine is effective.

In consideration of the fact that the values of these factors are rather difficult to determine with the help of direct experiments, in WAT a relatively simple method is used to determine divergent values on the basis of a suitable elaboration of oscillograph records of the run of propulsive force R (τ) and of pressure in the combustion chamber p (τ) obtained when the engine is running.

To evaluate these factors, besides empirical curves of the force and pressure suitable theoretical dependences were used. The concern here above all is for patterns for propulsive force R and unit impulse J_1.

$$R = \left[\varphi_1 \varphi_2 \frac{k_0}{g} F_w(k, \xi_a) + \xi_a^2 X_a - \xi_a^2 \frac{p_z}{p}\right] F_{\min} p \qquad (1)$$

$$J_1 = \frac{\varphi_1 \sqrt{\chi f_0}}{k_0} \left[\frac{k_0}{g} F_w(k, \xi_a) + \frac{\xi_a^2 X_a}{\varphi_1 \varphi_2} - \frac{\xi_a^2 p_z}{\varphi_1 \varphi_2 p}\right] \qquad (2)$$

where: f_o -- diminished force of prepellant

p -- pressure in combustion chamber

p_z -- external pressure

g -- acceleration of gravity

ξ_a -- coefficient of nozzle divergence of supersonic element

X_a -- relation of exhaust pressure p_a to pressure in combustion chamber

k_o -- size depending on the exponent of the adiabat k

$F_w(k, \xi_a)$ -- function of the coefficient ξ_a and exponent of the adiabat k

φ_1 -- coefficient of loss in velocity in nozzle flux

φ_2 -- coefficient of losses in gas output in the nozzle

χ -- coefficient of heat losses

On the basis of dependences (1) and (2) only 2 out of 3 quantities interesting us can be determined.

From the analysis of basic dependences in the theory of solid propellant rocket engines

-- the **equation** for the velocity of gas exhaust

$$W = \varphi_1 F_w(k, \xi_a) \sqrt{\chi f_0} \approx \varphi_1 \sqrt{\chi} F_w(k, \xi_a) \sqrt{f_0} \tag{3}$$

-- the expression for gas output

$$G = \frac{\varphi_2 k_0 F_{min} p}{\sqrt{\chi f_0}} = \frac{\varphi_1 \varphi_2 k_0 F_{min} p}{\varphi_1 \sqrt{\chi} \sqrt{f_0}} \tag{4}$$

-- the pattern for propulsive force

$$R = \varphi_1 \varphi_2 F_R(k, \xi_a) F_{min} p \quad (5)$$

-- the pattern for unit impulse

$$J_1 = \frac{\varphi_1 \sqrt{\chi} \sqrt{f_0}}{k_0} \left[F_R(k, \xi_a) - \frac{\xi_a^2 p_z}{\varphi_1 \varphi_2 p} \right] \quad (6)$$

-- the expression for charge parameter

$$K = \frac{\delta S A_1 \sqrt{\chi f_0}}{\varphi_2 k_0 F_{min}} = \frac{\varphi_1 \sqrt{\chi} \delta S A_1 \sqrt{f_0}}{\varphi_1 \varphi_2 k_0 F_{min}} \quad (7)$$

the result is that the factors φ_1, φ_2 and χ enter the proper eauations solely in the form of the products $\varphi_1 \cdot \varphi_2$ and $\varphi_1 \sqrt{\chi}$

Moreover when solving a series of practical problems knowledge of the values of these products should be sufficient.

The product $\varphi_1 \varphi_2$ can be determined from pattern (1) if it is disposed of by oscillograph records of pressure and force obtained at the same time during combustion of the propellant in the examined engine. However, the product $\varphi_1 \sqrt{\chi}$ can be determined from dp dependence (2), in which the unit impulse will be defined from area measurements of curve R (τ) for the burned mass of charge ω. A change in the character of gas flow during engine run to a small degree affects the value of divergent products, however, from a practical point of view it is useful to determine the average value of the product $\varphi_1 \cdot \varphi_2$ in the whole time interval of the engine run from 0 to τ_c.

Then according to transformation (1) we get

$$\varphi_1 \varphi_2 = \frac{g}{k_0 F_w(k, \xi_a)} \left[\frac{\int_0^{\tau_c} R(\tau)\,d\tau}{F_{min} \int_0^{\tau_c} p(\tau)\,d\tau} + \frac{\xi_a^2 X_a \tau_c}{\int_0^{\tau_c} p(\tau)\,d\tau} - \xi_a^2 X_a \right] \quad (8)$$

Conducting analogical divergences in support of dependence (2) we obtain an expression for the product in the form of

$$\varphi_1 \sqrt{\chi} = \frac{k_0}{\sqrt{f_0}} \frac{J_1}{\frac{k_0}{g} F_w(k, \xi_a) + \frac{\xi^2 X_a}{\varphi_1 \varphi_2} - \frac{\xi_a^2 X_a \tau_c}{\varphi_1 \varphi_2 \int_0^{\tau_c} p(\tau)\,d\tau}} \quad (9)$$

Solving the system of equations (8) and (9) with regard to the unknowns $\varphi_1 \varphi_2$ and $\varphi_1 \sqrt{X}$ the sought after values of the products can be determined. Of course the value of parameters k_o, ξ_a, $F_w (\xi_a, k)$, X_a, F_{min}, g, f_o, for given conditions of investigations are known, however the remaining quantities can be determined from the proper oscillograph records.

In several instances it is necessary to know not the products but the individual coefficients. Then the method given above for determining coefficients of gasodynamic and heat losses in the form of corresponding products is not suitable.

Keeping in mind the limited character of the method discussed a procedure is presented below which makes it possible to establish each of

the divergent quantities φ_1, φ_2 and

The tests conducted showed that the product $\varphi_1 \cdot \varphi_2$ determined for a definite pressure range is conditioned mainly by the shape of the nozzle. On the other hand the value of the product $\varphi_1 \sqrt{\chi}$ depends to a large extent on the type of charge and the manner of its combustion in the engine chamber.

Heat losses can be limited by combustion of the charge solely on the surface of its duct in a chamber having walls covered with a layer of thermal insulation. In this case the value of losses will be practical--negligible, and it can be accepted that the coefficient of heat losses is close to the unit $\sqrt{\chi} \cong 1$.

This makes it possible to determine the value of the coefficient φ_1 from expression (9) and then φ_2 from expression (8).

In many instances of combustion of propellant charges having various shapes and sizes there exists a need to define the value of the coefficient of heat losses in the time function of the engine run.

Knowledge of heat losses and hence of the changes in the coefficient of losses $\chi(\tau)$ is necessary among other things when solving the basic problem of ballistics of the internal srps.

To determine the coefficient of heat losses χ a method cab be used based on use of the equation of gas flux and oscillograph records p (τ) obtained by using a measuring device making it possible to interrupt combustion of the charge in an arbitrary moment.

The pattern for second output of gases can be written in the form

$$G = \frac{d\omega}{d\tau} = \frac{\varphi_2 k_0 F_{min} p}{\sqrt{\chi f_0}} \quad (10)$$

Integrating (10) within limits of from $\tau = 0$ to $\tau = \tau_i$ after conformal transformations we obtain the dependence for evaluating the coefficient of heat losses occurring in the chamber to the moment of extinguishing the charge τ_i

$$\chi = \left(\frac{\varphi_2 k_0 F_{min}}{\sqrt{f_0}\, \omega_i}\right)^2 \left(\int_0^{\tau_i} p(\tau)\, d\tau\right)^2 \quad (11)$$

where $\int_0^{\tau_i} p\, d\tau$ -- impulse of force of powder gases conforming to the moment when combustion is interrupted

f_0 -- diminished propellant force

φ_2 -- value of the coefficient of losses of expenditure defined for a given nozzle when $\chi = 1$, i.e., when heat losses are negligible

ω_i -- weight of burned charge to the τ_i moment defined as the difference of the weights of the beginning ω and remaining after being extinguished.

Executing a series of interrupted combustions in the whole running interval of the engine $(0, \tau_c)$, we can determine the values of the coefficient of heat losses χ_i conforming to this moment. After executing the diagram $\chi = f(_i)$ and approximation of the obtained curve by conforming analytical dependence we get the expression $\chi(\tau)$ relatively $\chi(\Psi)$ conforming to the given conditions of the charge and engine construction.

This expression can be used in the ballistic design of srps of a similar type.

The average value of the coefficient of heat losses $\chi_{\acute{s}r}$ for the entire running period of the engine can be determined fo from the pattern:

$$\chi_{\acute{s}r} = \left(\frac{\varphi_2 k_0 F_{min}}{\sqrt{f_0}\,\omega} \right)^2 \left(\int_0^{\tau_c} p(\tau) \, d\tau \right)^2 \qquad (12)$$

where $\int_0^{\tau_c} p \, d\tau$ -- total impulse of gas pressure in the combustion chamber

On the basis of preliminary tests conducted it can be proven that the method used not only enables the determination of the values of individual coefficients of gasodynamic and heat losses but also the establishment of optimal shape and sizes of the nozzle. Moreover, the method makes it possible to investigate the effect on the heat losses of various parameters of charging when using chambers whose walls are in this or another manner

shielded from the reaction of the flame and also for chambers with walls free of thermal insulation.

Conducting the above investigations in support of the method discussed can contribute to the elaboration of the most effective method of decreasing heat losses for conforming shapes of propellant charges when using various materials and thicknesses of the thermal insulation layer for combustion chambers of solid propellant rocket engines.

Paper published in November, 1974

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